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COVER SHEET FOR TECHNICAL MEMORANDUM

TITLE- Design and Performance of Small Nuclear Stages Compatible With Small and Intermediate Class Launch Vehicles

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(NASA-CR-103930) DESIGN AND PERFORMANCE OF SMALL NUCLEAR STAGES COMPATIBLE WITH SMALL AND INTERMEDIATE CLASS LAUNCH VEHICLES (Bellcomm, Inc.) 37 p

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a Class nuclear engine currently requires utilization of a Saturn V since sub-orbital start will in all likelihood be prohibited on early flights. Launch vehicle costs are therefore a major factor in overall development cost of flight hardware. If the nuclear engine and stage are reduced in size to the point where test articles could be flown on less expensive launch vehicles such as TIIIM, significant reductions in early test and technology development program cost would result. In this study conceptual designs of small nuclear stages compatible with TIIIM, and an intermediate launch vehicle in the 100,000 lb payload class are undertaken. Performance and application of such a stage to useful unmanned missions are evaluated presuming nuclear stage startup after earth orbit has been achieved.

A nuclear stage sized for TIIIM insertion to a 100 n.m. parking orbit is, within present uncertainties, only competitive with a suborbitally launched TIIIM/Centaur for velocities ranging from earth escape to solar system escape. Payloads are reduced by over 20% if 300 n.m. parking orbit insertion is required for range safety. However, substantial payloads (i.e., 10,500 lbs to Mars and 4,300 lbs to Jupiter) are still achievable. Nuclear stages compatible with intermediate 100,000 lb class boosters have slightly improved performance compared to advanced cryogenics of equivalent gross weight launched from 100 n.m. circular parking orbit. Range safety penalties for 300 n.m. parking orbit insertion result in approximately a 12% payload reduction.

It is concluded that without suborbital start, utilization of the TIIIM/Nuclear stage as an injection stage for unmanned missions cannot be justified solely on the basis of performance. Added program cost and complexity of missions utilizing a nuclear stage would have to be incurred on grounds of sustaining nuclear engine development.

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TECHNICAL MEMORANDUMI. INTRODUCTION

Development of a nuclear engine in the 75,000 lb_f Nerva thrust range currently requires utilization of a Saturn V launch vehicle for flight test and qualification since sub-orbital start will in all likelihood be prohibited on early flights. Vehicle costs are, as a consequence, a major factor in overall development cost of flight hardware. If the nuclear engine and stage were reduced in size to the point where test articles could be flown on less expensive launch vehicles, significant reductions in early test and technology development program cost would result.

The purpose of this study is to conceptually design a small nuclear stage compatible with a relatively cheap launch vehicle such as TIIIM, and evaluate its performance and application to useful unmanned missions.

II. MISSION APPLICATION OF SMALL NUCLEAR ENGINESManned Missions

Manned planetary missions demonstrate the firmest requirement for nuclear rocket propulsion, and have provided the basic impetus for nuclear engine development. Substantial weight savings compared to cryogenics (on the order of 50% of initial weight in earth orbit for some missions) can result.

For orbiter and landing missions, low altitude circular planetary capture orbits are ideal for purposes of gathering high quality data. However, adoption of less ambitious data goals permits use of highly elliptic capture orbits at great overall weight savings. As noted in Reference 1, thrust in the range from 10,000 lb_f to 40,000 lb_f would be applicable for planetary capture and escape maneuvers if elliptical parking orbits were utilized.

This thrust range can also be employed for escape from highly elliptic earth orbit (slightly less than escape velocity) using small injection stages matched to Saturn V payload insertion capability. This mode is somewhat less efficient than launch from low circular earth orbit since nuclear stage performance is utilized over a smaller impulsive velocity range. However, a major new engine development could be eliminated.

Possible earth orbital and lunar applications for engines smaller than Nerva are discussed in Reference 2.

Unmanned Missions

The results of this study will show that a single 10,000 lb_f thrust engine could be flown on a small nuclear stage compatible with TIIIM, and would be capable of launching unmanned planetary payloads of more than 10,000 lbs to Mars and 500 lbs to solar escape. Performance would only approximate that of advanced cryogenics. However a small nuclear stage used for unmanned missions could concomitantly serve as a test and qualification vehicle for man rated engines ultimately to follow. The degree of added program cost and complexity for unmanned missions might conceivably be justifiable if, as a result, nuclear rocket development were sustained. Ultimately a single engine size could suffice for both manned and unmanned mission areas.

III. CURRENT STUDY OBJECTIVES

Success of the small engine/small stage development approach is in great measure contingent on achieving acceptable nuclear stage performance for the high energy unmanned probe missions. Here stage performance is extremely sensitive to mass fraction (ratio of propellant weight to gross stage weight). To gain a realistic measure of small nuclear stage performance a preliminary stage design analysis is undertaken. TIIIM and an uprated Titan (Reference 3) with a large diameter core and 156 inch solids (TIIIG) are assumed as earth orbit injection vehicles. Scaling laws are derived for nuclear stages sized for both launch vehicles. Velocities in the range from earth escape to greater than solar system escape are considered. Comparisons with competitive cryogenic vehicles are made to gauge relative performance capability. A "rubberized" stage design is assumed in which the propellant container, residuals, payload shroud, etc., are varied as a function of propellant and payload weights. No restart capability is assumed, precluding use for synchronous orbit insertion. Comments about the implications of restart are, however, included in ensuing discussions.

This study was limited to startup of the nuclear stage after earth orbit has been achieved, as will undoubtedly be the case in initial flights. If suborbital start can be employed at a later time, performance would exceed that determined here.

IV. RANGE SAFETY

Range safety constraints have a significant impact on nuclear stage performance. In view of present AEC and NASA policy uncertainties, several representative launch modes are considered.

Factors governing nuclear rocket range safety constraints are functionally grouped in two categories: pre-start and post-start. Hazards associated with nuclear reactors are due primarily to fission product radiation emitted during operation of the reactor. Prior to operation the only radiation source is uranium with low activity. During and after operation at high power, the fission product radiation is orders of magnitude more intense than the uranium radiation, but unlike uranium radiation will decay appreciably with time. One or a few years is generally considered necessary for fission products to decay to a tolerable level.

Pre-Start

During ground handling and launch, sub-criticality (i.e., no sustained neutron chain reaction possible) must be insured for all occurrences, including destruction of the launch vehicle. Satisfactory systems have been devised for guaranteeing launch pad reactor sub-criticality without interference with engine operation.

Post-Start

An engine failure shortly after startup could fix the radioactive stage in low earth orbit. If the orbit decayed more rapidly than the fission product radioactivity, random dispersion of radioactive material could result during atmosphere reentry. The nuclear engine might disassemble breaking into small inconspicuous pieces of "hot" material. Small highly radioactive fragments and dust might result upon impact. (This combination of events presents the greatest hazard to man and animals.)

A number of schemes have been considered to insure that such events will not occur:

1. The nuclear stage could be placed in a high, long-lived parking orbit, allowing sufficient time in event of failure for fission products to decay prior to atmospheric reentry.
2. A chemical Auxiliary Thrust System (ATS) could either de-orbit the stage safely (into the ocean) or boost the stage into a long-lived orbit (Reference 4). The problem here is insuring integrity of the vehicle after failure so that the ATS will have time to respond. An emergency cooling system is required for this purpose. Because of weight penalties and system complexity, practical utilization of an ATS must be considered speculative at this time.
3. The use of multiply redundant systems could insure nuclear engine insertion into high parking orbit before complete failure. Current thinking at NASA and AEC suggests that almost all penalizing safety constraints could ultimately be eliminated by this means, in which case selected design improvements and flight experience could eventually make low orbit insertion (or possibly suborbital operation) of nuclear stages acceptable. (Suborbital start would result in substantial increases in payload, generally 20 to 50%.)

V. INSERTION ORBIT SELECTION

Nuclear stage insertion to 100 n.m. is taken as nominal, representative of safety mode 3. Parking orbit lifetime is from 2 to 5 days. If the stage were to disassemble during ignition individual fuel elements would reenter within a day.

A parking orbit of 300 n.m. is presumed to satisfy range safety requirements via mode 1. This would provide a lifetime of approximately 5 years for an intact stage devoid of propellants. If the reactor disassembled after failure, individual fuel elements would enter considerably sooner, between one and two years.*

*As a rough estimate, an orbital lifetime of one or two years should allow fission products to decay to a level safe for random reentry.

VI. LAUNCH VEHICLE CONSTRAINTS

TIIM launch capability direct to 100 n.m. circular orbit is approximately 38,000 lbs. The TIIM second stage is not restartable and as a result payload direct to higher circular orbit degrades rapidly. Direct launch payload to 300 n.m. circular orbit is reduced by 20% to 30,500 lbs. To achieve a greater payload in 300 n.m. circular parking orbit, the stage can be launched to a 300 n.m. x 100 n.m. ellipse and circularized with a solid auxiliary propulsion unit (jettisonable along with the nuclear stage fairing and shroud prior to stage ignition). Gross payload to elliptical orbit is 35,500 lbs, and auxiliary propulsion is 1,500 lbs (approximately 4% of gross weight) so that net payload to 300 n.m. parking orbit is 34,000 lbs. (An alternate means of achieving 300 n.m. orbit is to circularize with the transtage, in lieu of the solid engine. This has not been examined in depth but does appear to be a satisfactory alternative.)

The TIIM/Nuclear configuration is shown in Figure 1. Stage diameter is 180 inches. The stage geometry is within limits for TIIM bulbous payloads so that no problems are posed for the range of payloads considered. (For reference TIIM bulbous payload capability is shown in Figure 2. Vehicle capability is greater than the 100% design wind allowable over the range of payload configurations considered.)

TIIG payload to 100 n.m. is approximately 100,000 lbs. Payload to 300 n.m. is estimated at 93,500 lbs, with solid motor circularization similar to TIIM. A 260 inch nuclear stage diameter, similar to the SIVB stage is selected so that the stage would be compatible with the Saturn vehicles as well. Since the TIIG has a 180 inch core this diameter would not represent an excessively bulbous payload.

Several points are worth noting about the interstage and fairing configuration chosen. These structures are of course jettisoned before nuclear stage startup. Since gross weight to orbit is fixed, the net penalty is a reduction in propellant weight to orbit (and minor reduction in stage inert weight commensurate with propellant reduction). Sensitivity of stage performance to weight jettisoned in earth orbit is 270 fps per percent of gross weight (independent of gross weight and impulsive velocity). For a 34,000 lb vehicle, a savings of 1% or 340 lbs, equivalent to a 4 ft shortening of adapter length, would therefore have only a minor effect on stage performance. Thus exotic means of reducing the interstage length by retracting the engine bell or even launching the stage upside down (engine forward) are not deemed worthwhile.

Payload density is assumed to be 6.5 lbs/ft^3 (similar to Surveyor) which is comparable with the density of liquid hydrogen propellant (4.4 lbs/ft^3). Stage and payload length based on the payload density estimate is therefore insensitive to payload variation, and a length to diameter ratio of approximately 4 prevails over the range of payloads for both TIIIM and TIIIG.

VII. DESIGN APPROACH

Gross stage weight (stage weight plus payload) is determined by gross weight to desired insertion orbit, minus jettisoned shroud, interstage, and auxiliary propulsion for parking orbit circularization (if required). Propellant weight to desired hyperbolic excess velocity is computed by integration of the equations of motion to include gravity losses (Reference 5). Inerts (comprised of engine, structure, systems and propellant residuals) are determined on the basis of gross stage and propellant weight. Discretionary payload is then equal to gross weight less inerts and propellant.

VIII. VEHICLE DESCRIPTION

The general Titan TIIIM/Nuclear stage configuration is shown in Figure 3. The stage is approximately 700 inches long and has a basic diameter of 180 inches. The stage is supported on Titan IIIM by a faired adapter mated to the 120 inch diameter core vehicle, forming a bulbous or hammerhead configuration. A jettisonable fairing is at the forward end. Modifications to this configuration for the 300 n.m. orbit are shown in Figure 4. The TIIIG/Nuclear stage is 260 inches in diameter and approximately 960 inches long, mated to a 180 inch core in similar fashion to the TIIIM configuration.

Propellant is LH_2 , loaded subcooled to limit propellant boiloff losses from radiation heating. The engine is graphite with zirconium-hydride for additional moderation, based on a design study by Aero-Jet General (Reference 6). Specific impulse is assumed at 850 sec (Reference 7).

Thrust for TIIIM and TIIIG versions are $10,000 \text{ lb}_f$ and $20,000 \text{ lb}_f$, respectively. Engine burn times vary from 900 to 3000 secs as a function of propellant loading.

Stage scaling laws including assumed engine weights for the range of propellant loadings considered were derived

from point designs and are given below. (Weight breakdowns for the point designs are discussed in Section X.)

LV	Gross Stage Weight	Inerts (W_s) Scaling Law
TIIM	$\sim 34,000$	$W_s = 6,350 + .168 W_p$ (including 3,500 lb engine)
TIIG	$\sim 92,000$	$W_s = 11,000 + .122 W_p$ (including 5,000 lb engine)

The graph in Figure 5 shows the TIIM point designs calculated for different propellant loadings and notes the degree of linearity over the propellant range. Figure 6 shows the relationship of payload, propellant, and inerts as a function of hyperbolic excess velocity. Note in particular the relationship of propellant to payload. Since gross stage weight is fixed, payload is increased only at the expense of reduced propellant. As a result, payload/velocity relationships differ considerably from those of fixed propellant stages. Mass fractions (ratio of propellant weight to propellant plus inerts weight) and payload fraction (ratio of payload to gross stage weight) are given as a function of propellant weight in Figure 7. (The sharp decrease in mass fractions for large payloads underscores the payload/propellant relationship cited above.)

IX. PERFORMANCE

TIIM/Nuclear payload for selected missions are presented in Figure 8 for both nuclear and competitive cryogenic stages. Figure 9 and 10 show relative performances for payloads from 200 to 15,000 lbs, over the range of velocities from earth escape to solar system escape. Differences in payloads inserted from 100 n.m. and 300 n.m. parking orbit dramatize

the penalties associated with nuclear stage range safety constraints. Figure 9 compares the rubberized nuclear stage with TIIIM/Centaur (and, at higher velocities, TIIIM/Centaur/Burner) launched optimally via direct ascent, and Figure 10 compares the nuclear stage with a hypothetical advanced cryogenic stage ($I_s = 460$ sec, $.90 < \lambda < .92$) launched from 100 n.m. parking orbit. This launch mode is not optimal for the advanced cryogenic stage, but does serve to indicate a minimum capability, and is suitable for purposes of comparison.

The 100 n.m. launched nuclear stage shows marginal payload improvement compared to Centaur for $V_c > 40,000$ fps. (This advantage, however, would be offset with addition of a Burner II fourth stage on Centaur.) The 300 n.m. vehicle is not competitive with Centaur over any range. The advanced cryogenic stage even if fired orbitally rather than by optimal suborbital staging, would be equal or superior to the nuclear stages over the entire range if the highest mass fraction (.92) were achievable.

Gravity losses for the TIIIM nuclear stage are shown for reference in Figure 11 for the 100 n.m. launch case with initial thrust to weight (T/W_i) = .292. Velocity losses vary from 250 fps at earth escape to 3,000 fps at solar system escape. Gravity losses for the 300 n.m. stage are almost identical.

TIIIG/Nuclear payloads range from almost 40,000 lbs to Mars to 2,000 lbs to solar escape. Performance is compared with an advanced 100,000 lbs cryogenic stage in Figures 12 and 13. Relative performance of the nuclear stage to the cryogenics is improved, principally as a result of reduced engine thrust-weight (T/W_e) in the 20,000 lb_f thrust range (as discussed in Section X). In overall performance, the 300 n.m. stage is directly competitive with the cryogenic stage at $\lambda = .90$, and the 100 n.m. stage is superior to all cryogenics over almost the entire velocity range. Increased payload margin is, however, only 10% which could be offset by direct launch of the cryogenic stage or addition of a Burner II stage.

X. DESIGN ELEMENTS

Radiation Environment

Payload and engine components are susceptible to damage from fission product radiation. In addition, propellant

boiloff and shielding penalties can result from radiation heating during long duration burns. These situations are assessed below:

- Payload - Electronic equipment can be designed so that cumulative doses of 10^6 rads of gamma rays and 10^{12} neutrons per cm^2 of fast neutrons will not degrade performance. As shown in Figure 14 (Reference 8) separation distance between engine and payload is sufficiently great in the present case to preclude payload shielding requirements over the range of propellant weights selected. (Note: possible design penalties associated with radiation hardening of electronic equipment to above values have not been considered.)
- Engine Components - Components adjacent to the engine are assumed radiation hardened or locally shielded. (300 lbs of shielding is included in the engine weight.) When possible, components are mounted forward of the propellant tank in the equipment bay area.
- Propellant Heating - With thermal mixing, no boiloff weight penalty due to radiation heating results if propellants are subcooled. Figure 15 (Reference 8) shows the subcooling requirements for the range of propellants considered. Assumed tank vent pressure is 30 psia. Without subcooling significant boiloff and/or propellant shielding penalties can result, as cited in Reference 8.

Thrust Selection

The dependence of payload on the ratio of thrust to initial vehicle weight (T/W_1) is shown in Figure 16. Engine weight has been assumed to be proportional to thrust, with thrust in lb_f equal to four times engine weight in lb_m .

Figure 16 shows that for a hyperbolic excess velocity of 0.35 emos, the optimum T/W_1 is approximately 0.2, independent of the stage scaling law. For other velocities the optimum T/W_1 is also 0.2.

The estimated engine thrust to engine weight (T/W_e) relationships for nuclear rocket engines are shown in Figure 17. While the general relationship is fairly linear for thrusts in the range of 20,000 lb_f to 75,000 lb_f, there is a significant flattening of the curve below the thrust level of 20,000 lb_f. Consequently, for small stages compatible with the TIIIM, the optimum T/W_i is increased because of the decrease in T/W_e . The 10,000 lb_f thrust engine of this study gives initial thrust to weights (T/W_i) of 0.292 and 0.334 for the 100 n.m. and 300 n.m. parking orbits, respectively. These T/W_i 's are considered to be close to optimum.

Substantial uncertainties in engine weight exist especially in the low thrust regions. These weights conceivably can be reduced by as much as 15% based on some projected advances in the state-of-the-art.

Structures

Structural components for the TIIIM nuclear stage are shown in an exploded view in Figure 18. Aluminum skins are used throughout.

The interstage adapter is a 260 in truncated cone. Auxiliary propulsion units are supported within this section as shown in Figure 4. The lower skirt cylinder is 26 in long and forms the support structure for the engine cone. The engine support is a truncated 45° cone, 70 inches in length.

The propellant tank is a welded aluminum monocoque structure designed to a limit pressure of 30 psia. The tank is comprised of a cylindrical center section, an elliptical upper dome, and a lower dome consisting of elliptical, conical, and spherical sections. The stage is "rubberized" by adding or removing cylindrical segments. No micrometeoroid protection is assumed.

The forward skirt is a 26 inch cylinder of fiber glass sandwich construction. This section forms the thermal barrier between the tank and the equipment section. The equipment section is a cylinder 64 inches in length housing stage systems and radiation sensitive engine components. The nose fairing is a conic section with a half cone angle of 17°. Length is varied with payload size by addition of a cylindrical extension.

The insulation arrangement is shown in Figure 19. Insulation is polyurethane foam bonded to the external surface of the tank, (essentially the same as the Centaur LH₂ tank).

Systems and Residuals Weight

Systems weight estimates are based on the advanced Centaur system in Reference 9. Centaur is of similar gross weight to the TIIIM/Nuclear stage and performs the same type of mission. Such items as pressurization and venting are scaled to nuclear stage LH₂ tank size. Penalties associated with bipropellant systems such as propellant utilization are discarded. Propellant residuals are estimated at 1.5% of propellant weight. Almost the entire residual loss is attributable to H₂ vapor at 30 psia tank pressure. Liquid residuals are negligible by comparison.

Weight Breakdown

A TIIIM/Nuclear stage weight breakdown for payload delivered to a hyperbolic excess velocity of .30 emos is given in Figure 20. (Structure and systems weights are designated in greater detail in Figure 21.) Structure and engine weights are each about 40% of total stage inerts. Systems weight is approximately 20%. Mass fraction and payload fraction are .667 and .157, respectively.

XI. POTENTIAL PERFORMANCE IMPROVEMENTS OF NUCLEAR STAGES

Significant improvements of TIIIM/Nuclear stage performance can potentially be achieved by the following means:

- Suborbital Start - Although it is unlikely that sub-orbital start would be permitted in initial flights, it could become accepted operational procedure later in the program. The performance of a TIIIM nuclear stage has not yet been evaluated for suborbital start. However, the performance potential is illustrated by comparable data for a Saturn V nuclear stage. Based on Nerva I performance, a stage optimized for sub-orbital start would increase payloads from 20% to 50% as compared with orbital start in the range of V_c from 36,000 to 60,000 fps, as shown in Figure 22. Injection into a lower altitude parking orbit, e.g. 80 n.m. parking orbit, followed by orbital start might alternately be acceptable. Propellant weight to orbit would in this case be increased by about 2%.

- Stage Weight Reduction - If 1,000 lbs of inerts were saved through engine, structure, and systems weight reduction, 1,000 lbs of additional payload would result. Uncertainties in nuclear engine weight at the 10,000 lbs level, alone, could account for as much as 500 lbs. In the current design conventional structural materials are utilized. Recourse to exotic materials, coupled with possible reduced propellant tank pressure requirements might save hundreds of pounds more. Flight systems weights, which are estimated on the basis of Centaur systems might also be reduced substantially. Stage performance reflecting a 1,000 lb inerts weight reduction is shown in Figure 23.
- Restart/Reduction of Gravity Losses - Restart capability would be desirable in conjunction with sub-orbital start in order to avoid the short launch windows associated with direct injection vs. parking orbit. It would also provide the capability for injection into earth synchronous orbit. Restartability can offer performance improvements for planetary mission via multiple burn injection. Figure 24 shows the gravity loss reduction for a two burn injection assuming various cooldown propellant losses. The first burn is to a one day ellipse with 100 n.m. perigee, and the second burn is started at an optimum point before perigee. If, for example, cooldown losses could be held to 3% of burned propellant weight, significant net velocity savings for V_c greater than 40,000 fps result. For Jupiter missions the savings would be about 800 fps. For solar system escape, saving would be on the order of 1400 fps.

If no additional inerts resulted for restart, payloads to Jupiter and solar system escape would be increased by 600 lbs and 800 lbs, respectively. Additional shielding, insulation, and engine penalties would be subtracted from these totals, so that net payload gain is uncertain.

XII. CONCLUSIONS

Without suborbital start, utilization of a TIIIM/Nuclear stage as a deep space injection stage for unmanned missions cannot be justified solely on a basis of performance.

A nuclear stage sized for TIIIM insertion to a 100 n.m. parking orbit is, within present uncertainties, only competitive performance-wise with a suborbitally launched TIIIM/Centaur, or advanced cryogenic stages launched from 100 n.m. parking orbit. Performance of TIIIM/Nuclear sized for 300 n.m. parking orbit insertion is reduced by greater than 20% compared to the 100 n.m. case and is considerably below that of TIIIM/Centaur although substantial payloads are still achievable.

The relative performance of TIIIG/Nuclear stages compared to cryogenics of similar gross weight is somewhat improved, notably for the 300 n.m. parking orbit case, but in absolute terms nuclear stage performance capability is only marginally superior to that of the cryogenics.

If assumed ground rules pertaining to restrictions of nuclear stage suborbital start are eased, performance would considerably exceed that determined here. Possible nuclear stage weight reductions based on advances in the state-of-the-art of engine, structures, and systems could also have a significant impact on stage performance, especially in the high energy range for $V_c > 46,000$ fps.

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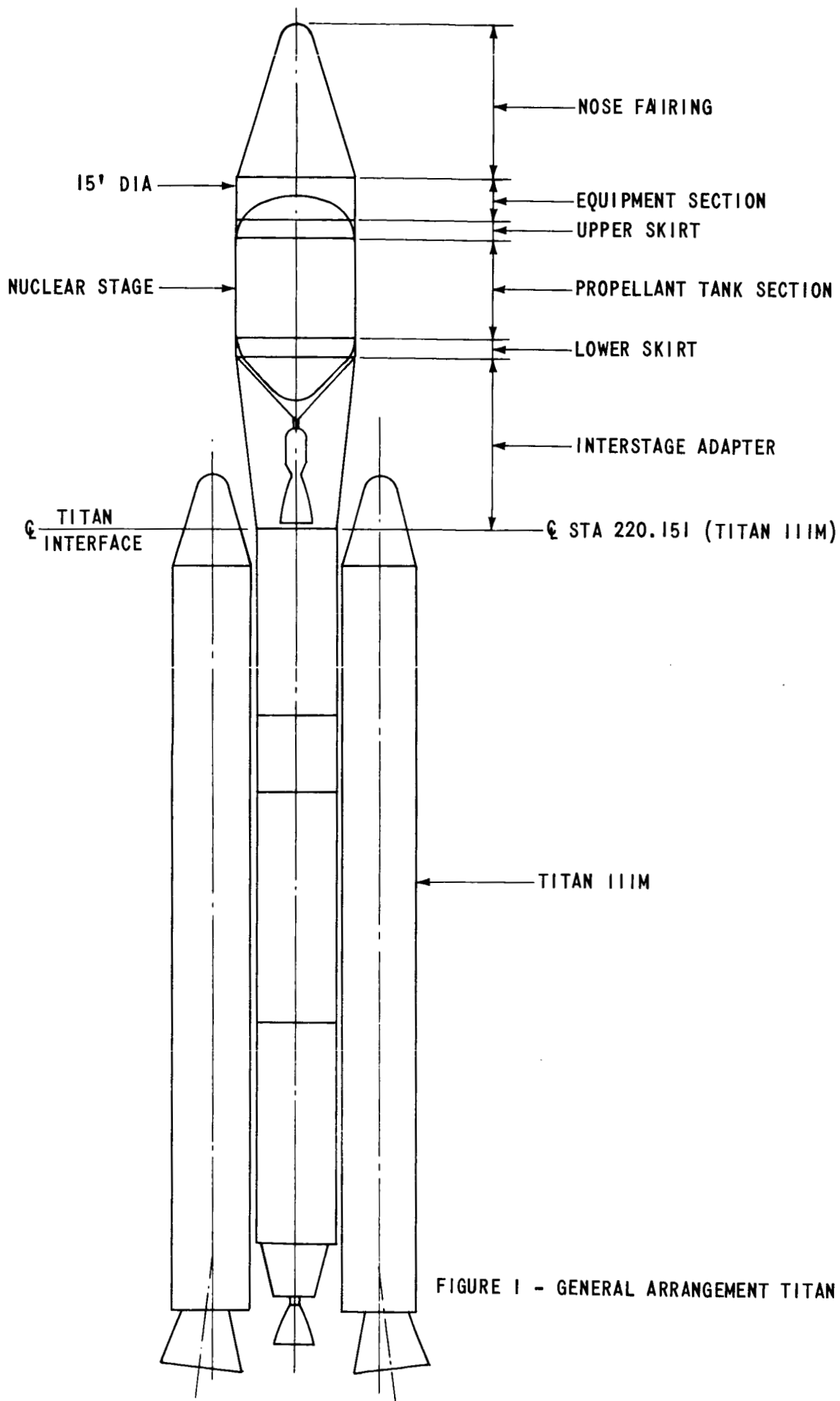
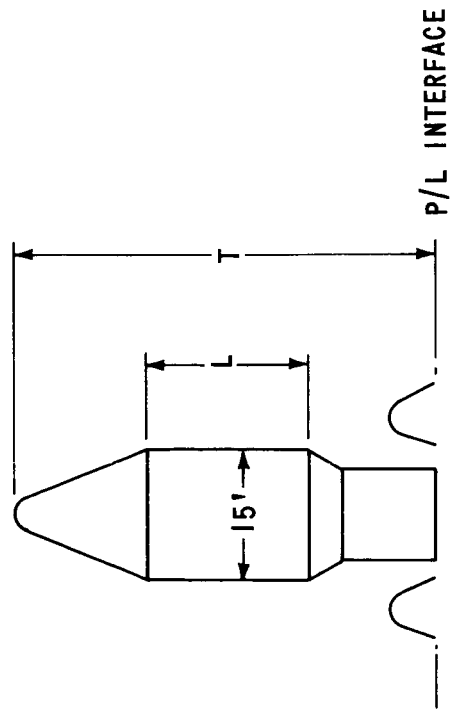
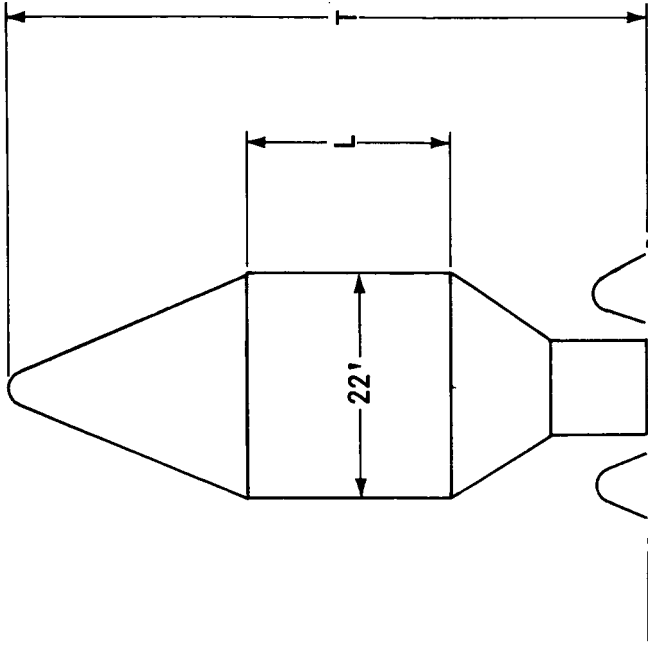


FIGURE 1 - GENERAL ARRANGEMENT TITAN IIIM/NUCLEAR

REF: MARTIN MARIETTA CORP



"L"	"T"	VEHICLE CAPABILITY & DESIGN WIND	
		MANNED	UNMANNED
16'6"	43'	148%	198%
23'6"	50'	111%	146%
33'6"	60'	75%	101%
43'6"	70'	56%	77%



"L"	"T"	VEHICLE CAPABILITY & DESIGN WIND	
		MANNED	UNMANNED
24'2"	63'6"	28%	43%
30'8"	70'	17%	27%

FIGURE 2 - TITAN III M BULBOUS CAPABILITY

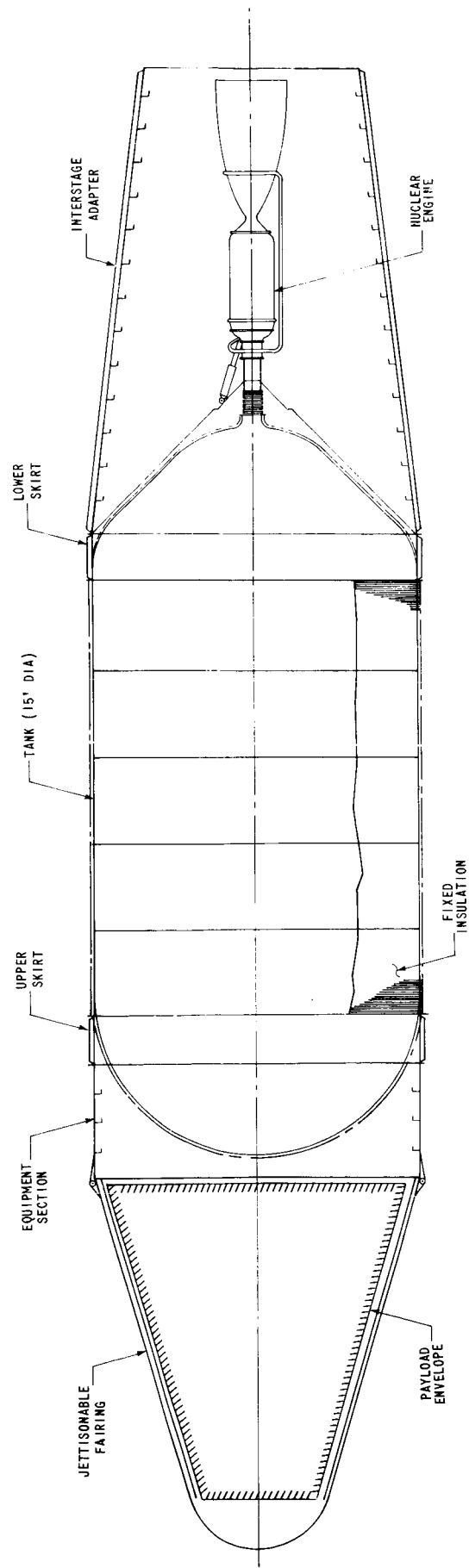


FIGURE 3 - NUCLEAR STAGE - 100 N.M. ORBIT CONFIGURATION

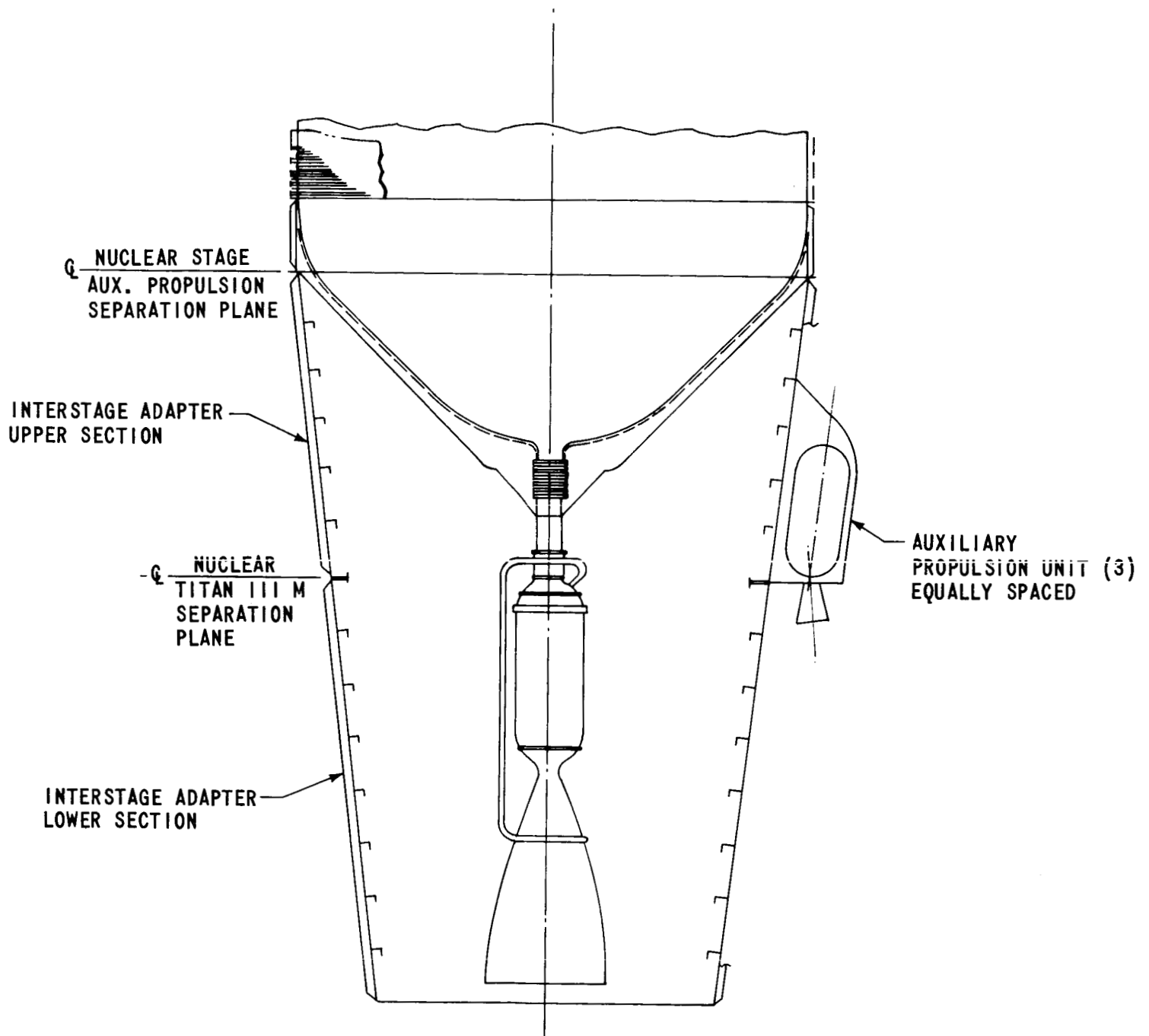


FIGURE 4 - NUCLEAR STAGE MODIFICATION 300 N.M. ORBIT

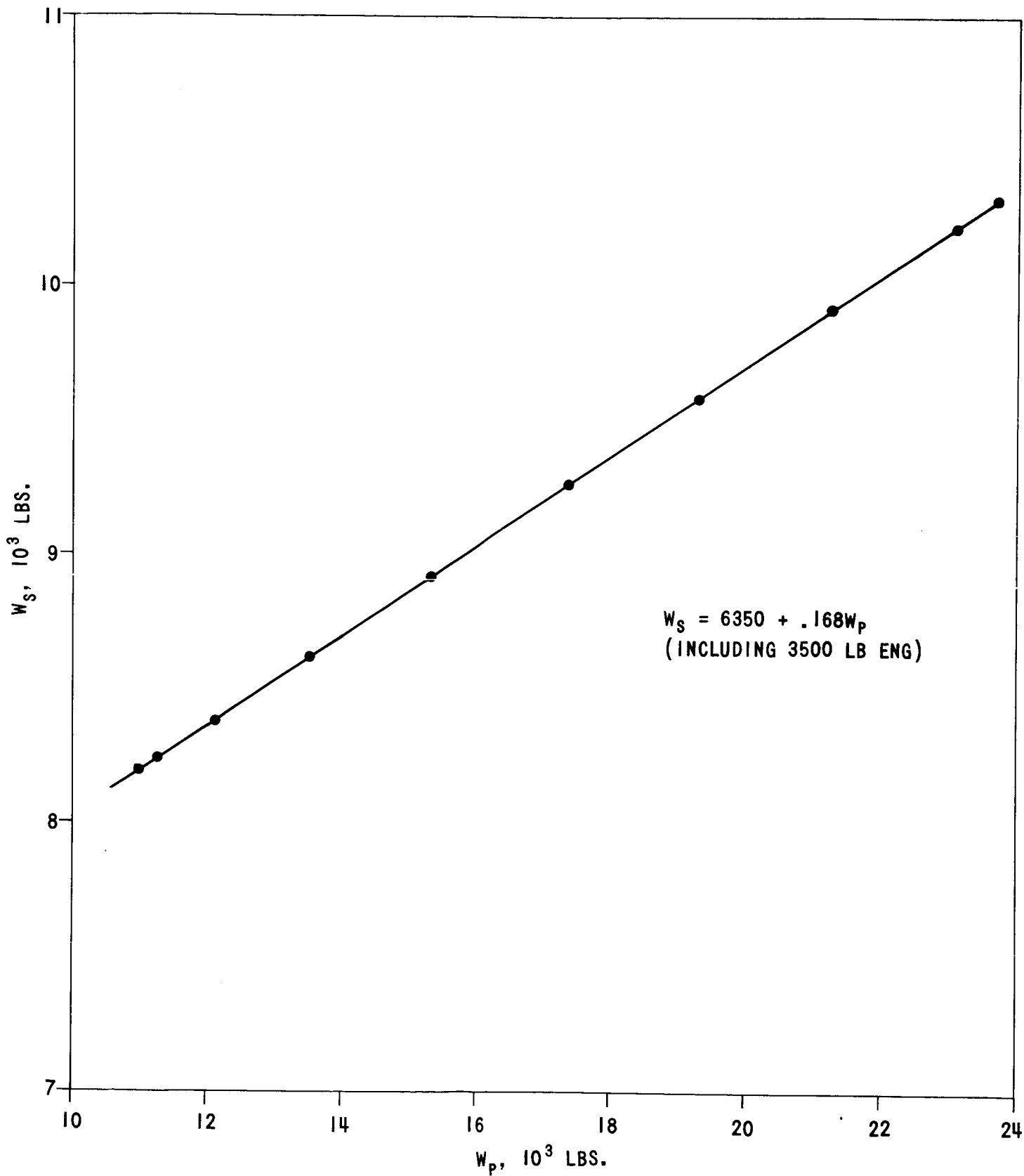


FIGURE 5 - T III M/NUCLEAR SCALING LAW

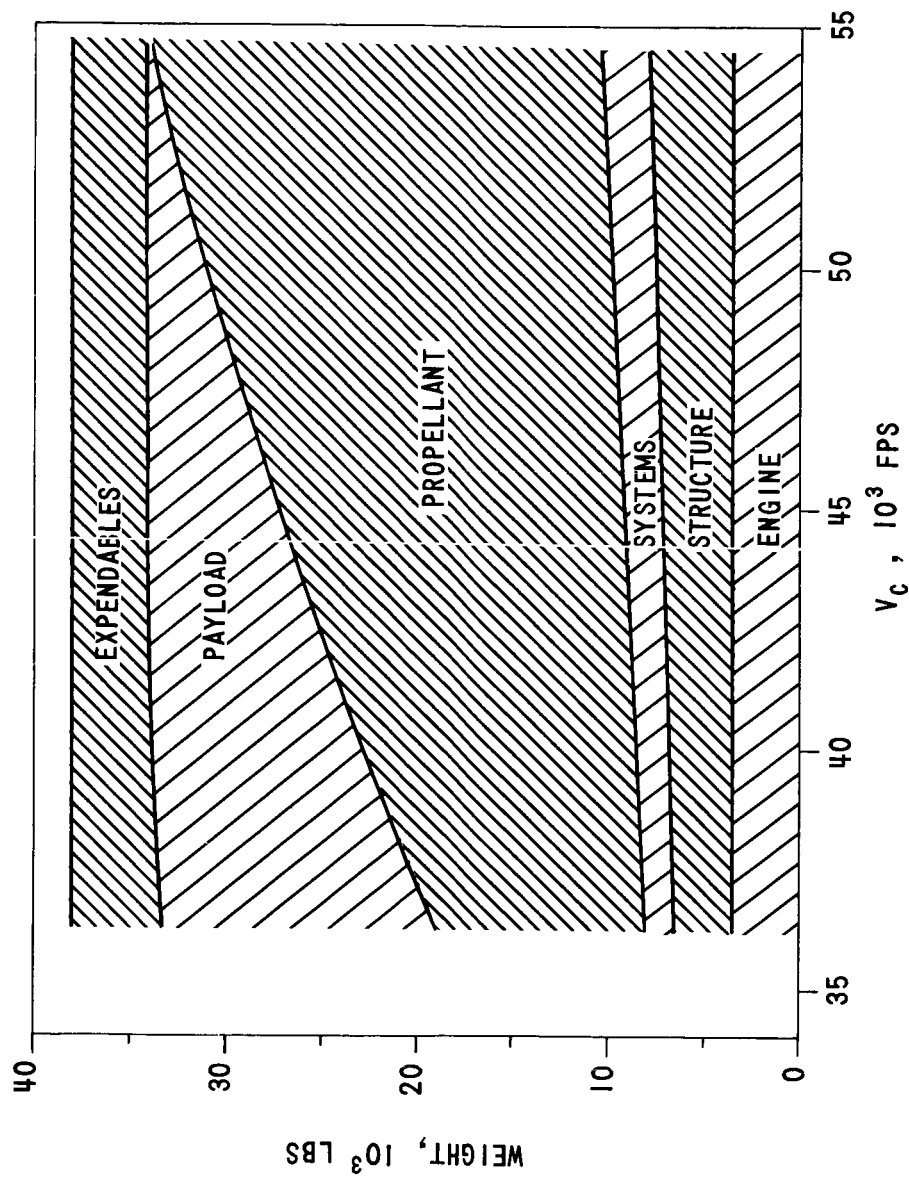


FIGURE 6 - T 111 M/NUCLEAR STAGE WEIGHT BREAKDOWN FOR
RANGE OF VELOCITIES

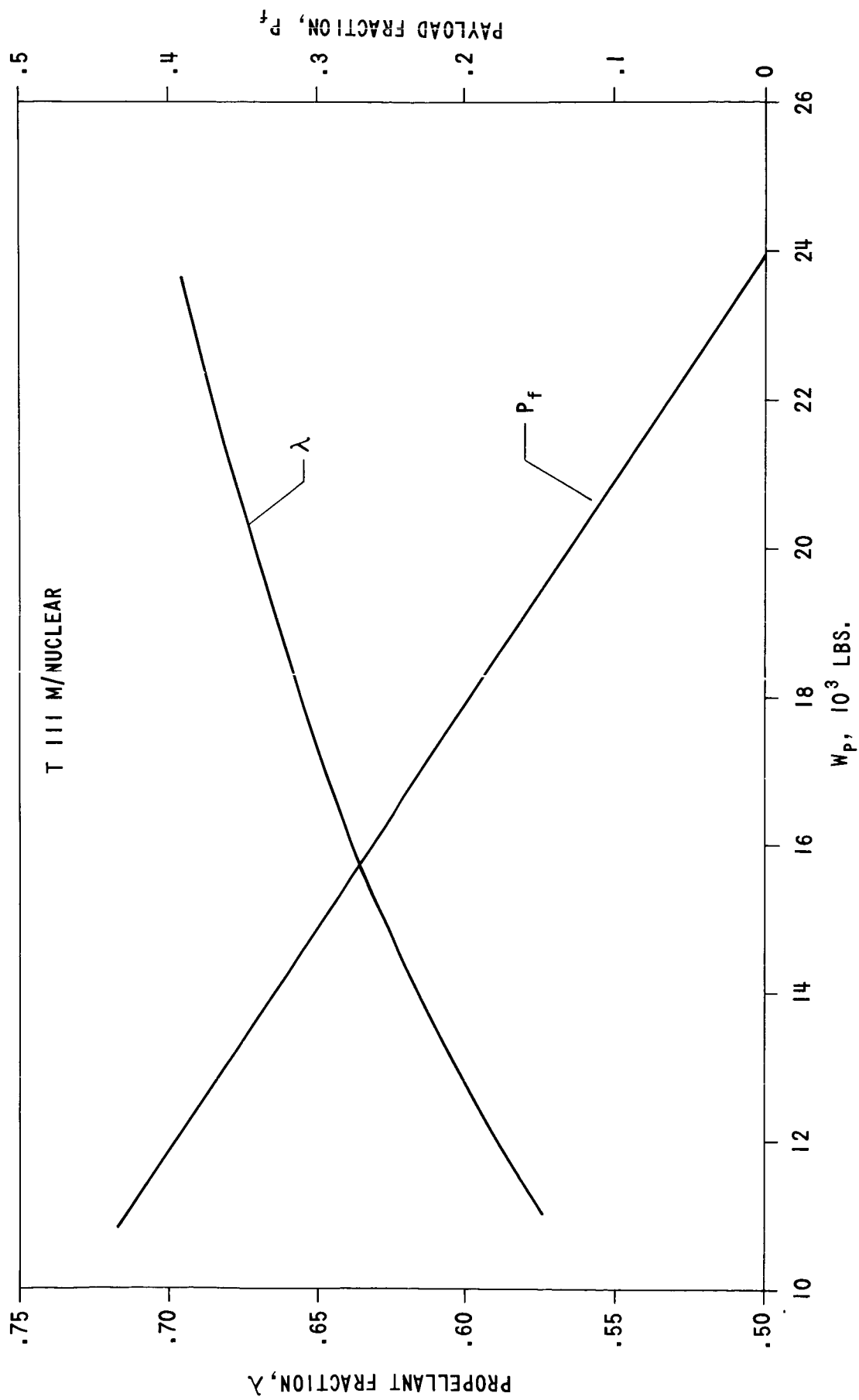


FIGURE 7 - PROPELLANT FRACTION AND PAYLOAD FRACTION VERSUS PROPELLANT

STAGE MISSION	V_c	V_∞	TIIM/NUCLEAR		TIIF CENTAUR (DIRECT)	TIIM ADVANCED CRYOGENIC (100 N.M. PARKING ORBIT)
			100 N.M.	300 N.M.		
MARS	37,800	0.113	12,800	10,530	13,320	12,670 - 13,220
VENUS	37,600	0.106	12,980	10,700	13,570	12,910 - 13,460
MERCURY (VENUS SWINGBY)	38,600	0.138	12,050	9,850	12,240	11,780 - 12,310
JUPITER	46,100	0.293	5,700	4,270	4,750	5,580 - 6,100
SATURN (JUPITER FLYBY)	46,600	0.301	5,320	3,970	4,420	5,250 - 5,820
SOLAR SYSTEM ESCAPE	54,000	0.410	620	—	?	1,540 - 2,300

FIGURE 8 - TIIM/NUCLEAR PAYLOAD FOR SELECTED MISSIONS

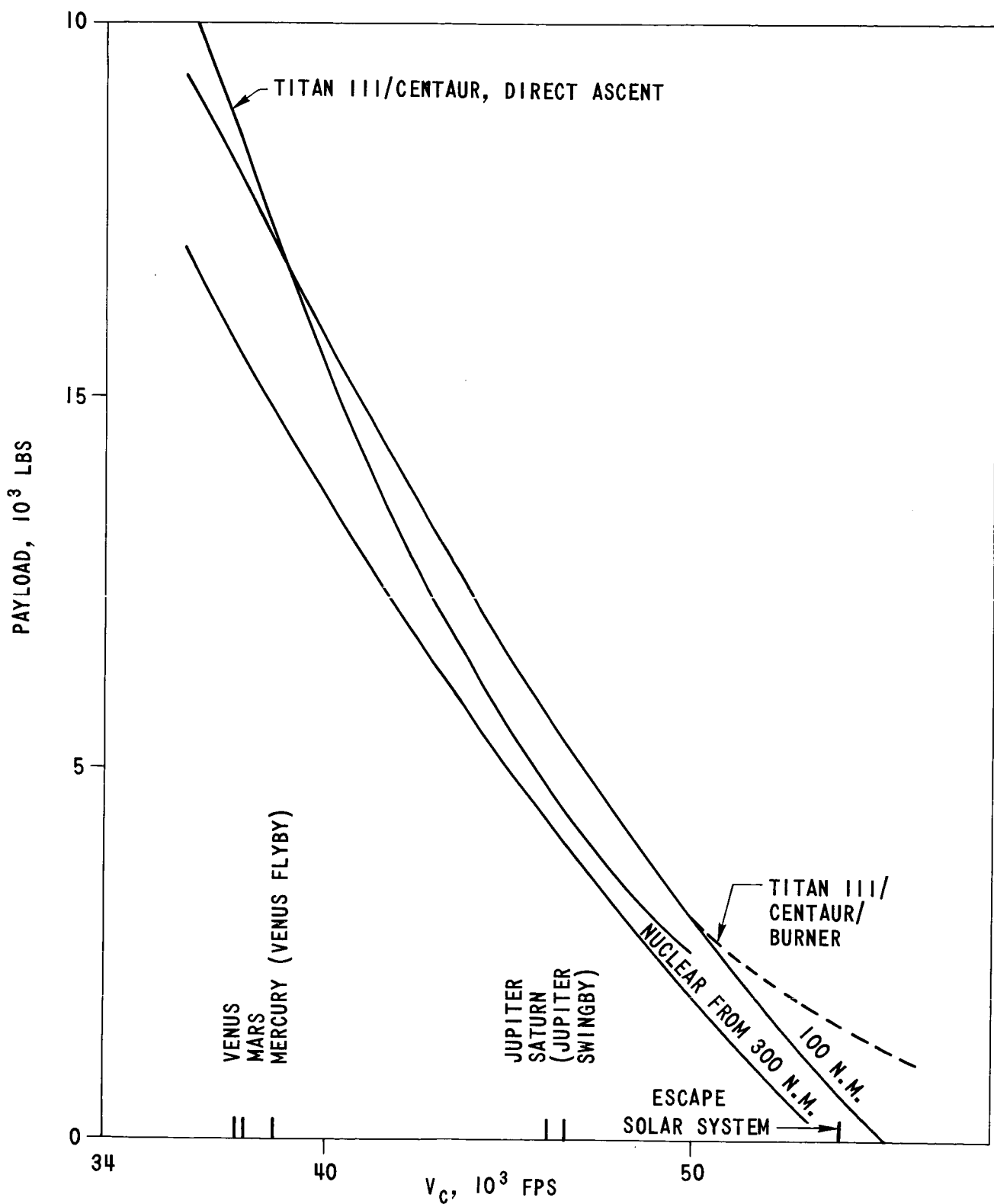


FIGURE 9 - COMPARISON OF T III M/NUCLEAR STAGE AND TITAN-CENTAUR PERFORMANCE

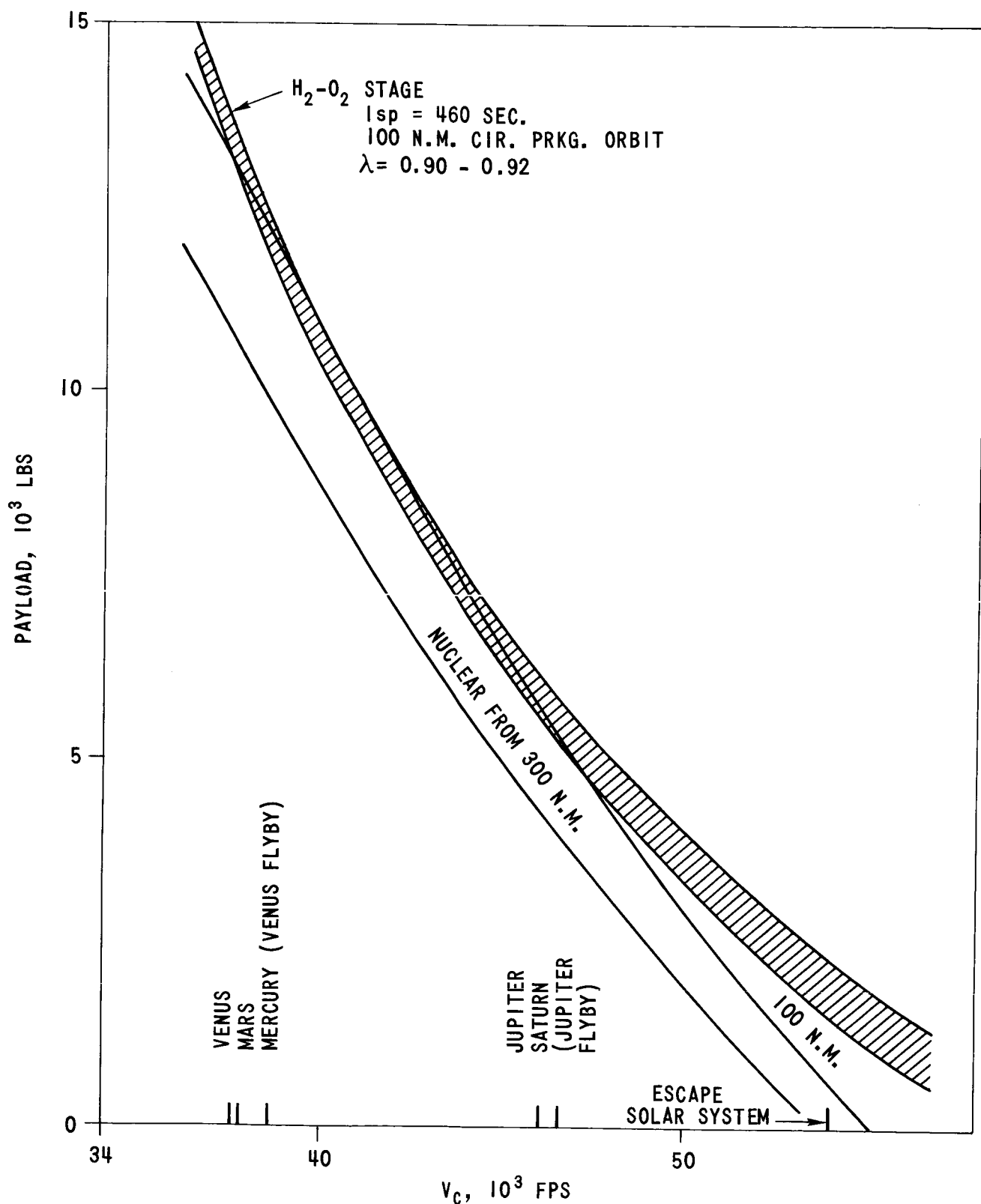


FIGURE 10 - COMPARISON OF T II M/NUCLEAR STAGE WITH ADVANCED H_2-O_2 STAGE (ORBITAL START)

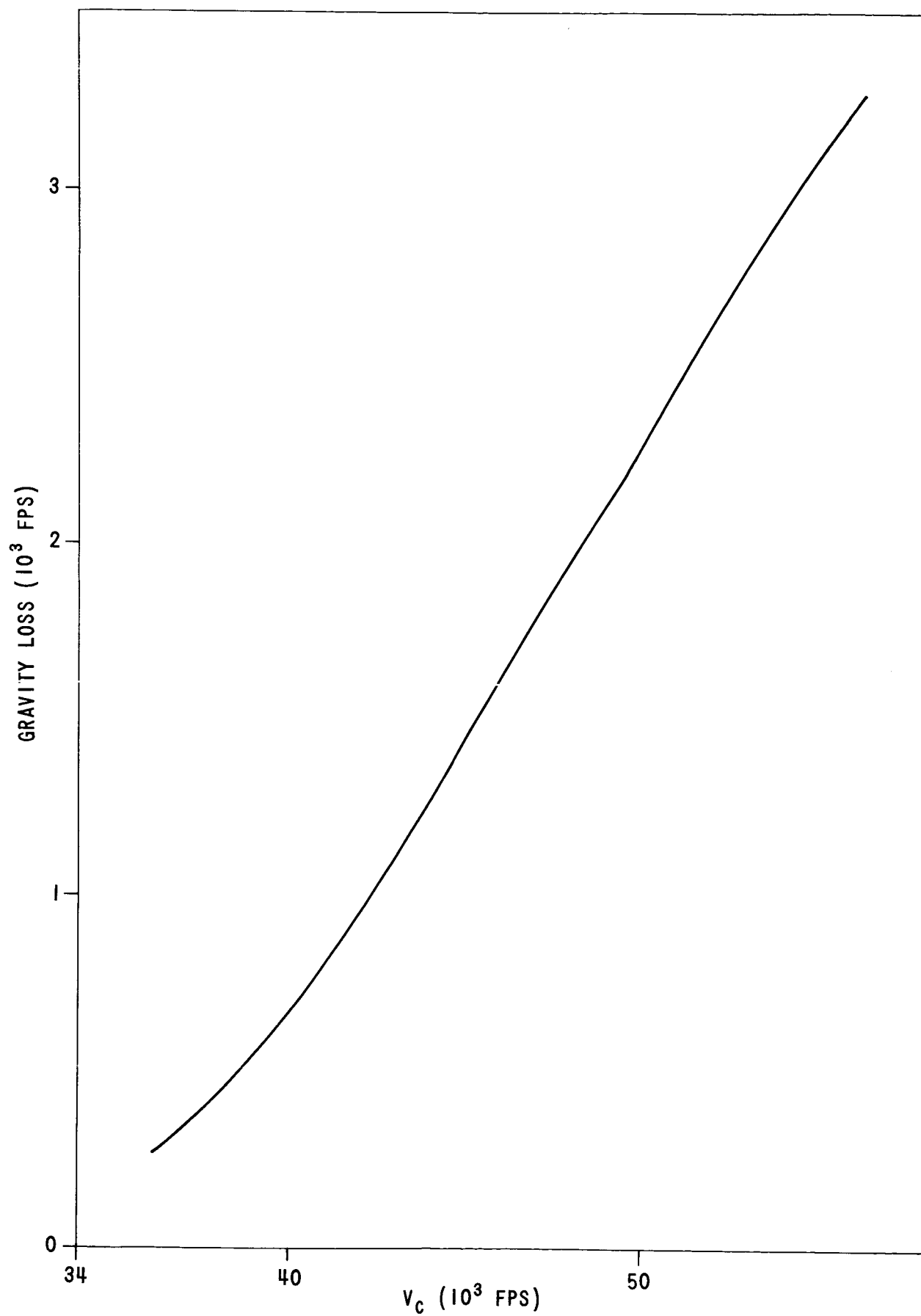


FIGURE 11 - GRAVITY LOSSES FOR T III M/NUCLEAR STAGE

STAGE MISSION	V_c	V_∞	TIIG/NUCLEAR		TIIM ADVANCED CRYOGENIC (100 N.M. PARKING ORBIT)
			100 N.M.	300 N.M.	
MARS	37,800	0.113	37,100	32,900	34,200 - 35,700
VENUS	37,600	0.106	37,600	33,300	34,900 - 36,300
MERUCRY (VENUS SWINGBY)	38,600	0.138	34,900	31,000	31,900 - 33,400
JUPITER	46,100	0.293	17,300	15,000	14,800 - 16,500
SATURN	46,600	0.301	16,400	14,100	13,900 - 15,700
SOLAR SYSTEM ESCAPE	54,000	0.410	4,000	3,000	4,100 - 6,000

FIGURE 12 - TIIG/NUCLEAR PAYLOAD FOR SELECTED MISSIONS

100,000 LB. IN ORBIT
 NUCLEAR ENGINE =
 20,000 LB. THRUST
 5,000 LB. WEIGHT

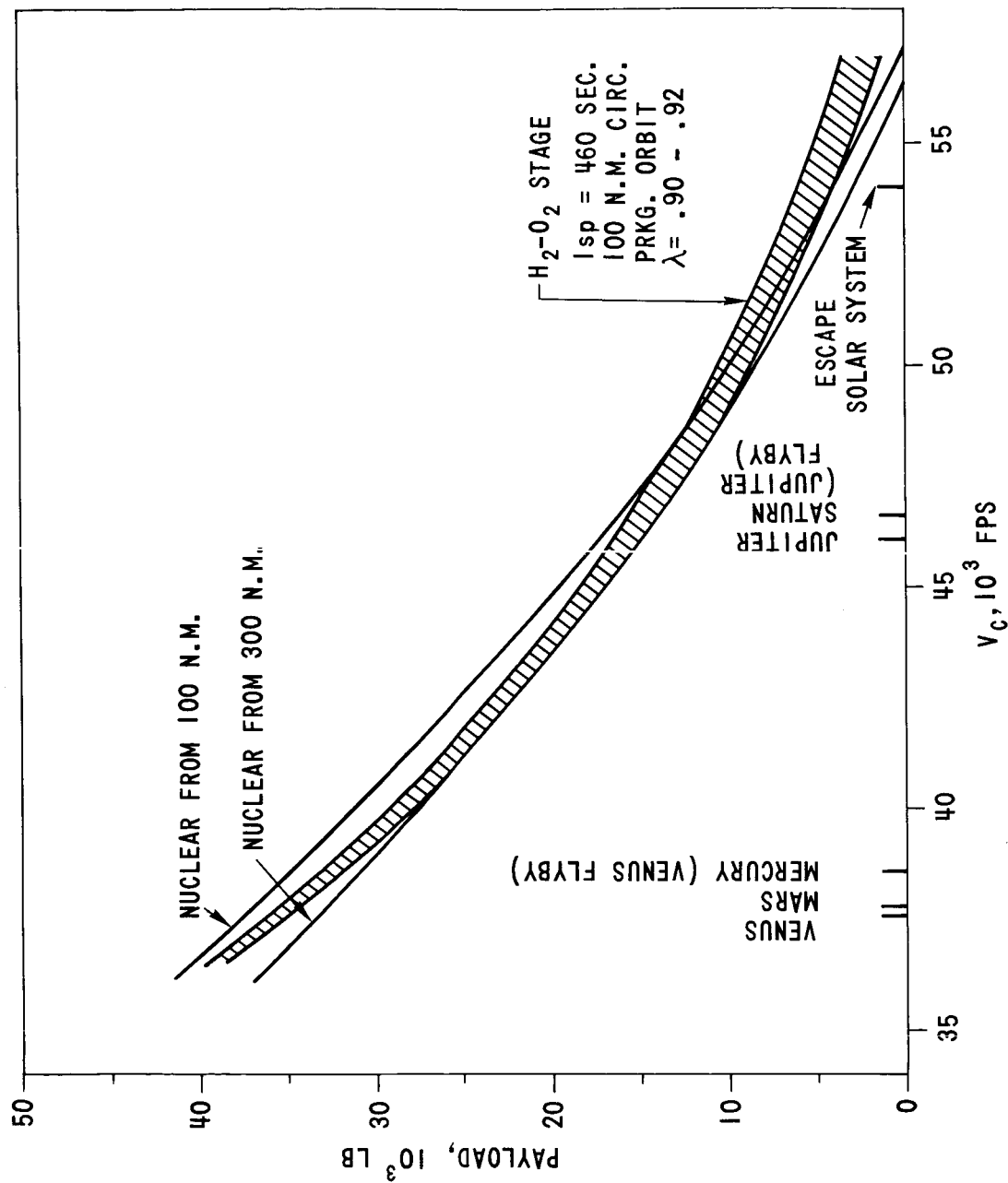


FIGURE 13 - COMPARISON OF T 111 G/NUCLEAR AND ADVANCED CHEMICAL PERFORMANCE (ORBITAL START)

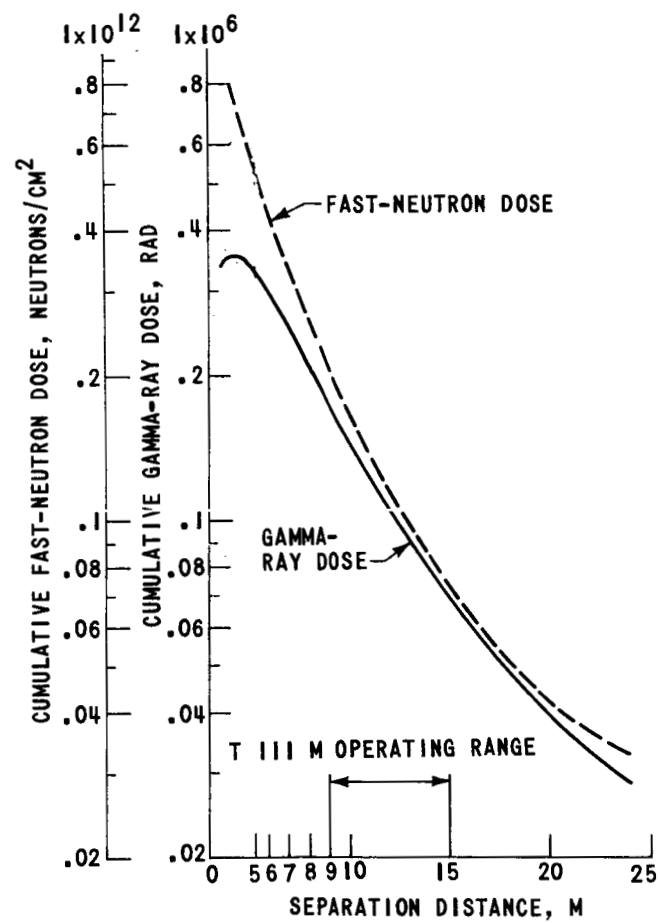


FIGURE 14 - CUMULATIVE GAMMA-RAY AND FAST-NEUTRON DOSE

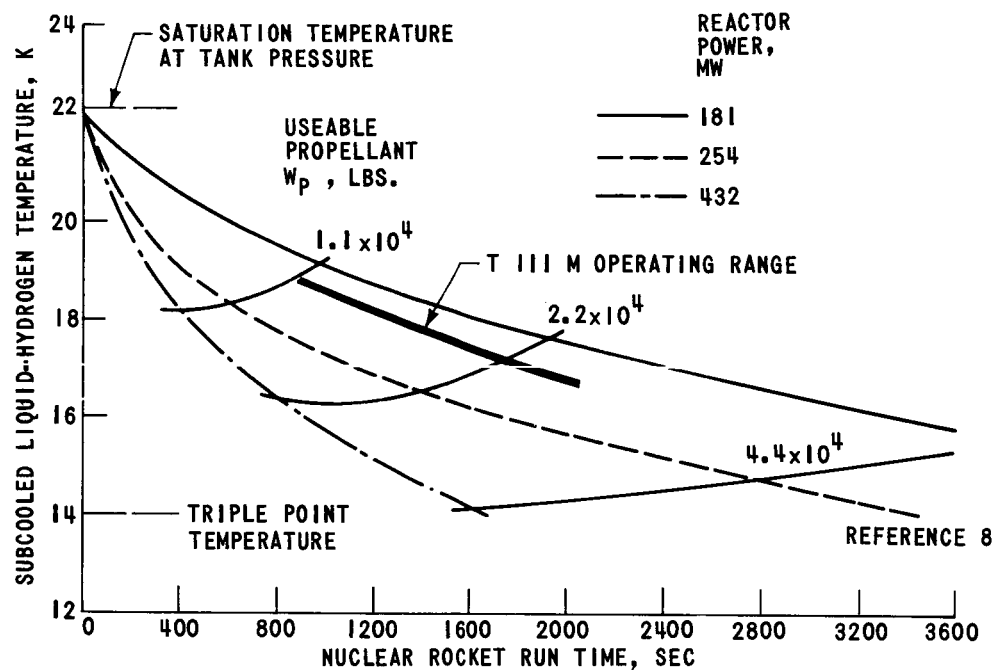


FIGURE 15 - PROPELLANT SUBCOOLING REQUIREMENT. ZERO NPSP AT END OF RUN TIME

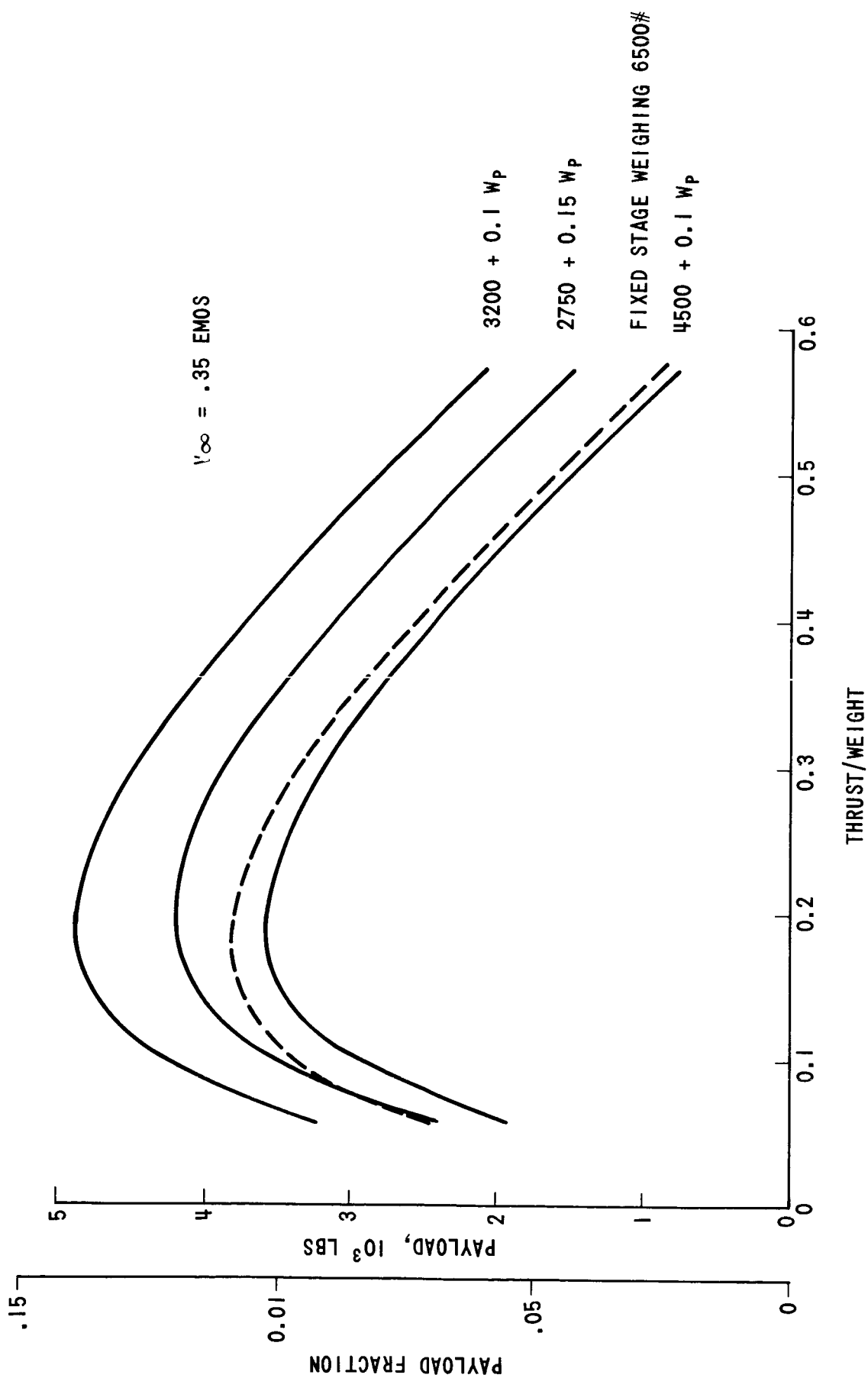


FIGURE 16 - DEPENDENCE OF PAYLOAD ON THRUST/GROSS WEIGHT

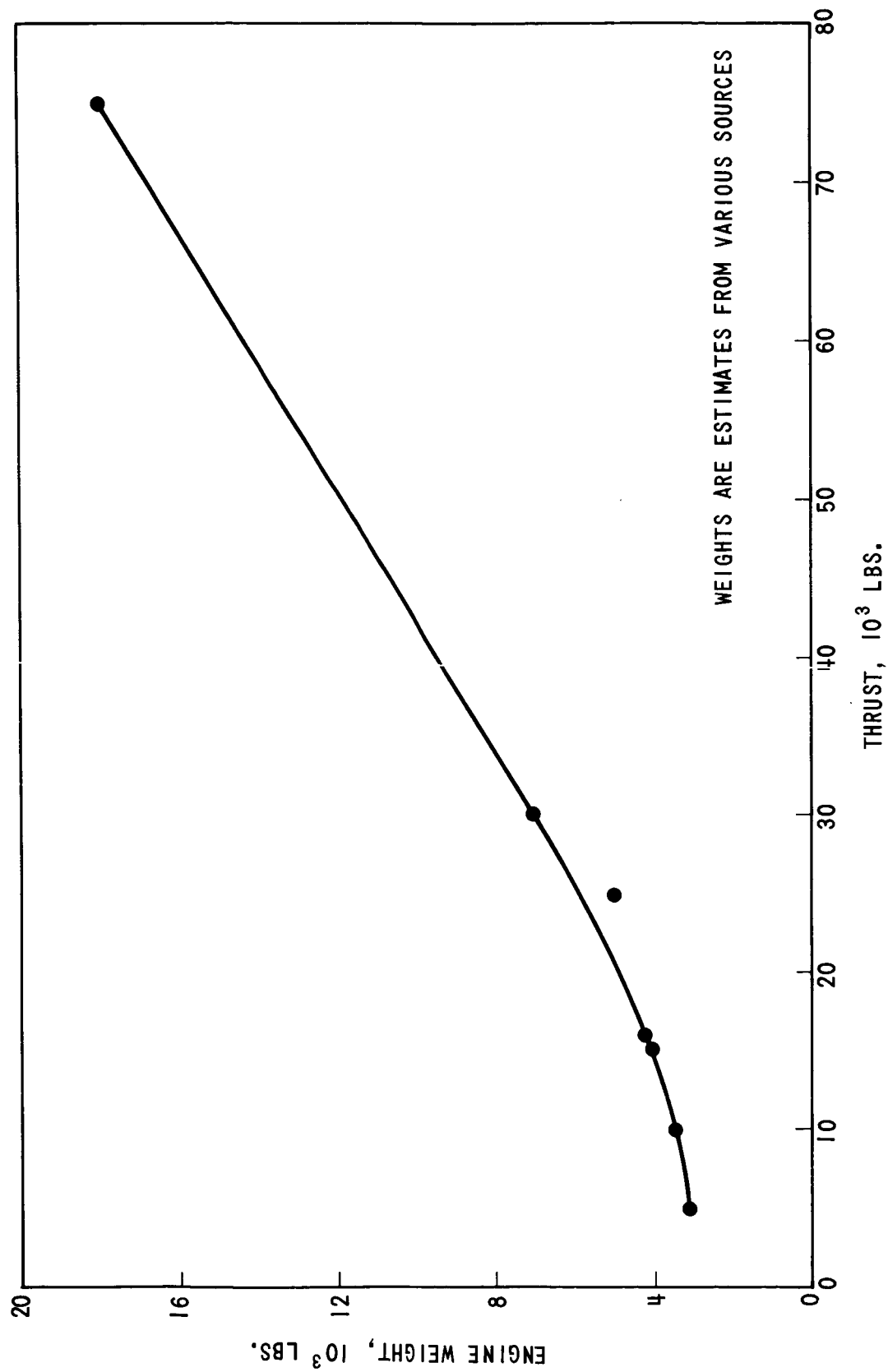


FIGURE 17 - NUCLEAR ENGINE WEIGHT vs. THRUST RELATIONSHIP

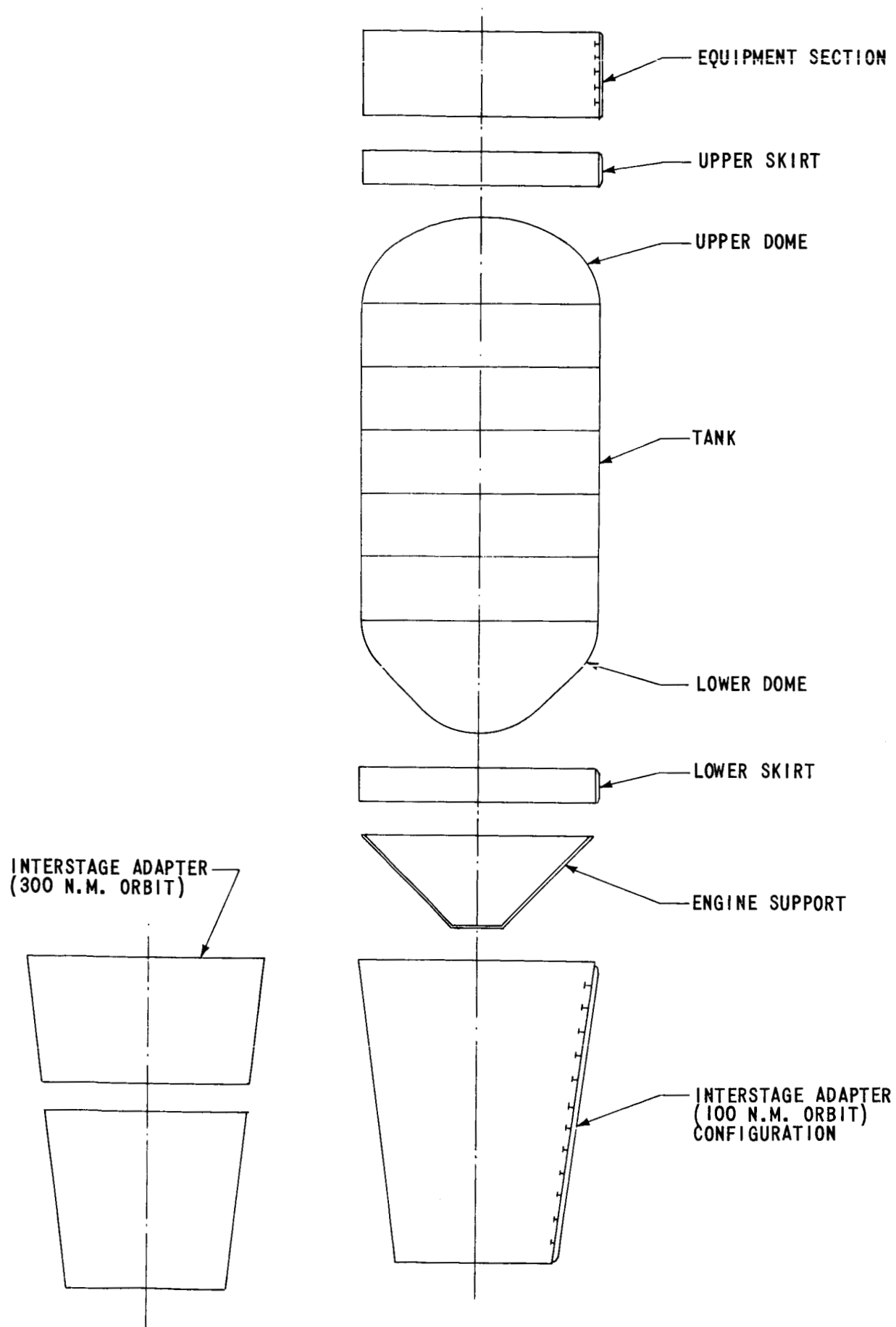


FIGURE 18 - NUCLEAR STAGE STRUCTURAL ARRANGEMENT

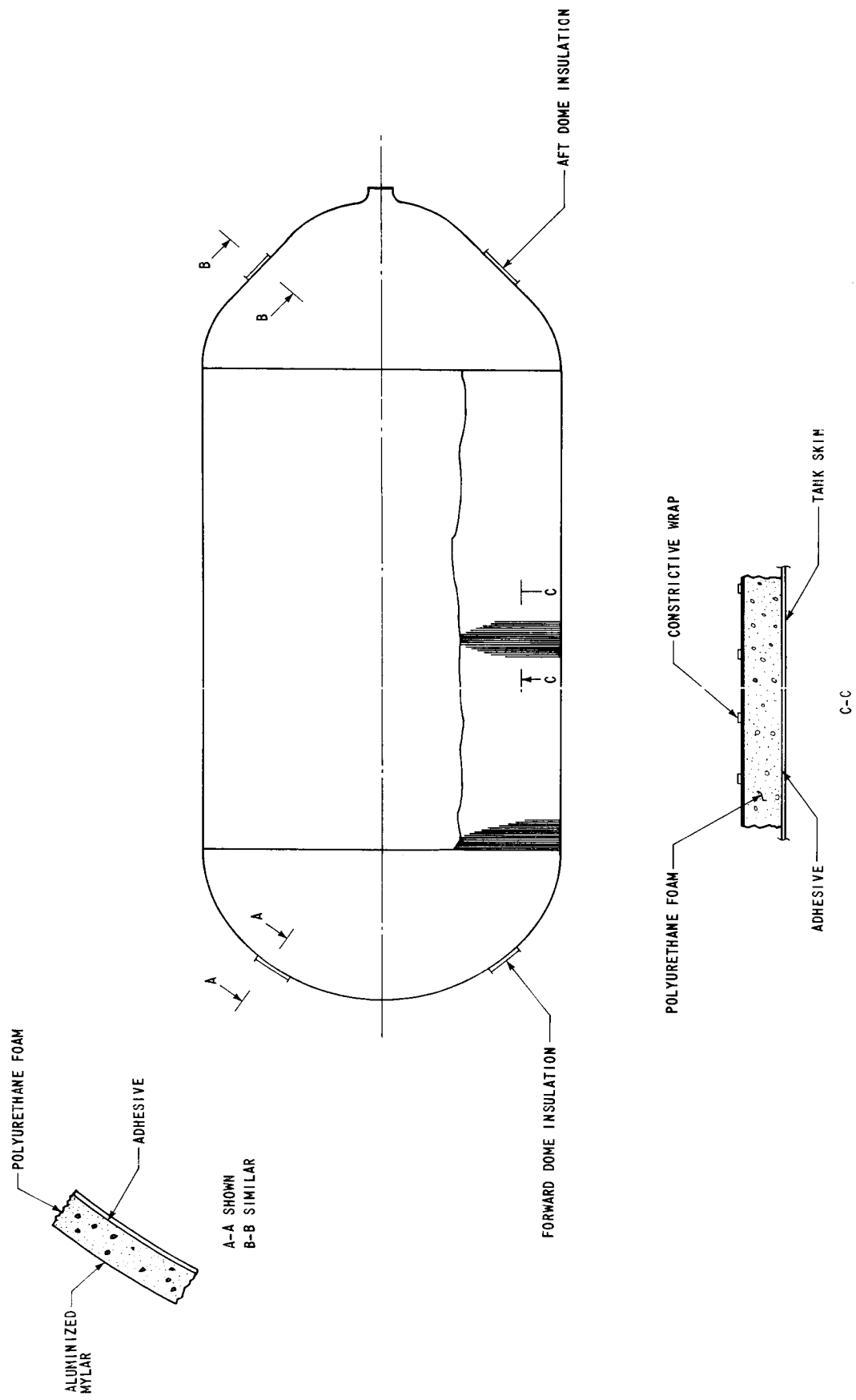


FIGURE 19 - INSULATION ARRANGEMENT PROPELLANT TANK

WEIGHT BREAKDOWN

EXAMPLE:

$$V_{\infty} = .30 \text{ EMOS}$$

GROSS WEIGHT FORWARD OF TIIM	38000
• EXPENDABLES	3793
FAIRING CONE	1250
ADAPTER	2543
	<hr/>
	34207
• INERTS AND NON CONSUMABLES	9301
STRUCTURE	3949
SYSTEMS	1852
ENGINE	3500
	<hr/>
	24906
• PROPELLANTS	19538
USABLE PROPELLANTS	19250
RESIDUALS	288
	<hr/>
	5368

$$\lambda (\text{USABLE PROPELLANT} / [\text{GROSS WEIGHT} - \text{PAYLOAD}]) = .667$$

$$\rho_f (\text{PAYLOAD} / \text{GROSS WEIGHT}) = .157$$

FIGURE 20 - TIIM/NUCLEAR DESIGN AND PERFORMANCE STUDY

WEIGHT BREAKDOWN - STRUCTURE

(LBS)

STRUCTURE	3091	
EQUIPMENT SECTION	550	
UPPER SKIRT	240	
TANKAGE	1641	
UPPER DOME		256
CYLINDER		1025
LOWER DOME		360
AFT SKIRT	330	
ENGINE SUPPORTS	190	
SECONDARY STRUCTURE	140	
INSULATION	670	
CONTINGENCY (5%)	188	
	<u>3949</u>	

WEIGHT BREAKDOWN - SYSTEMS

PROPULSION SYSTEMS	210
PROPELLANT LOADING	35
PRESSURE AND VENT	180
SECONDARY PROPELLANT	180
PNEUMATIC	80
SEPARATION	44
ELECTRICAL SUPPLY	335
GUIDANCE	425
CONTROL	50
TELEMETRY	150
TRACKING	25
RANGE SAFETY	50
CONTINGENCY (5%)	88
	<u>1852</u>

FIGURE 21 - TIIIM/NUCLEAR DETAILED WEIGHT BREAKDOWN

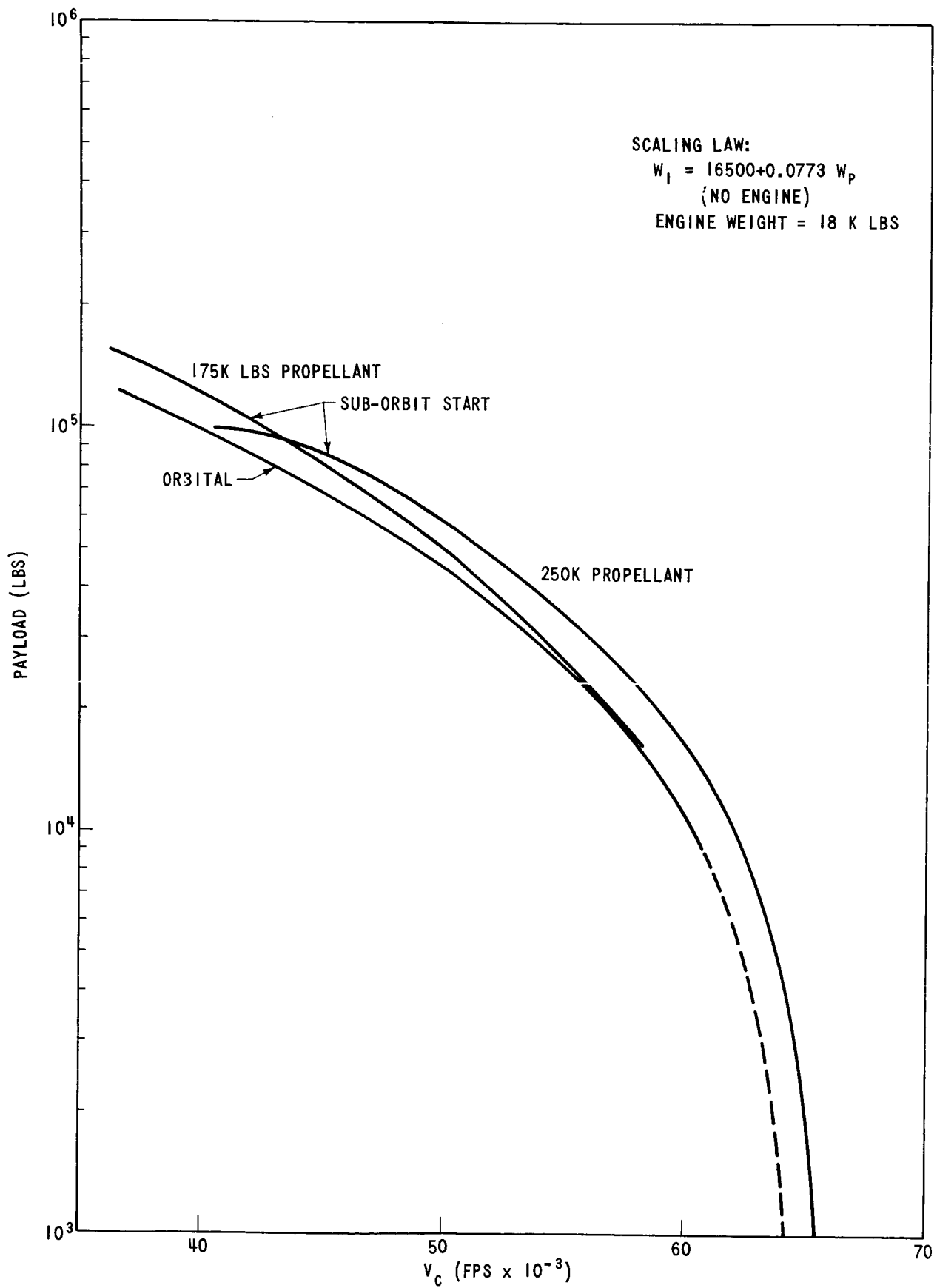


FIGURE 22 - NERVA SUBORBITAL START/ORBITAL START COMPARISON

STAGE MISSION	V_c	V_∞	TIIIM/NUCLEAR		TIIIF CENTAUR (DIRECT)	TIIIM ADVANCED CRYOGENIC (100 N.M. PARKING ORBIT)
			100 N.M.	300 N.M.		
MARS	37,800	0.113	13,800	11,530	13,320	12,670 - 13,220
VENUS	37,600	0.106	13,980	11,700	13,570	12,910 - 13,460
MERCURY (VENUS SWINGBY)	38,600	0.138	13,050	10,850	12,240	11,780 - 12,310
JUPITER	46,100	0.293	6,700	5,270	4,750	5,580 - 6,100
SATURN (JUPITER FLYBY)	46,600	0.301	6,320	4,970	4,420	5,250 - 5,820
SOLAR SYSTEM ESCAPE	54,000	0.410	1,620	900	?	1,540 - 2,300

FIGURE 23 - TIIIM/NUCLEAR OPTIMISTIC NUCLEAR STAGE PAYLOAD
FOR SELECTED MISSIONS (1000 LB INERTS WEIGHT
REDUCTION)

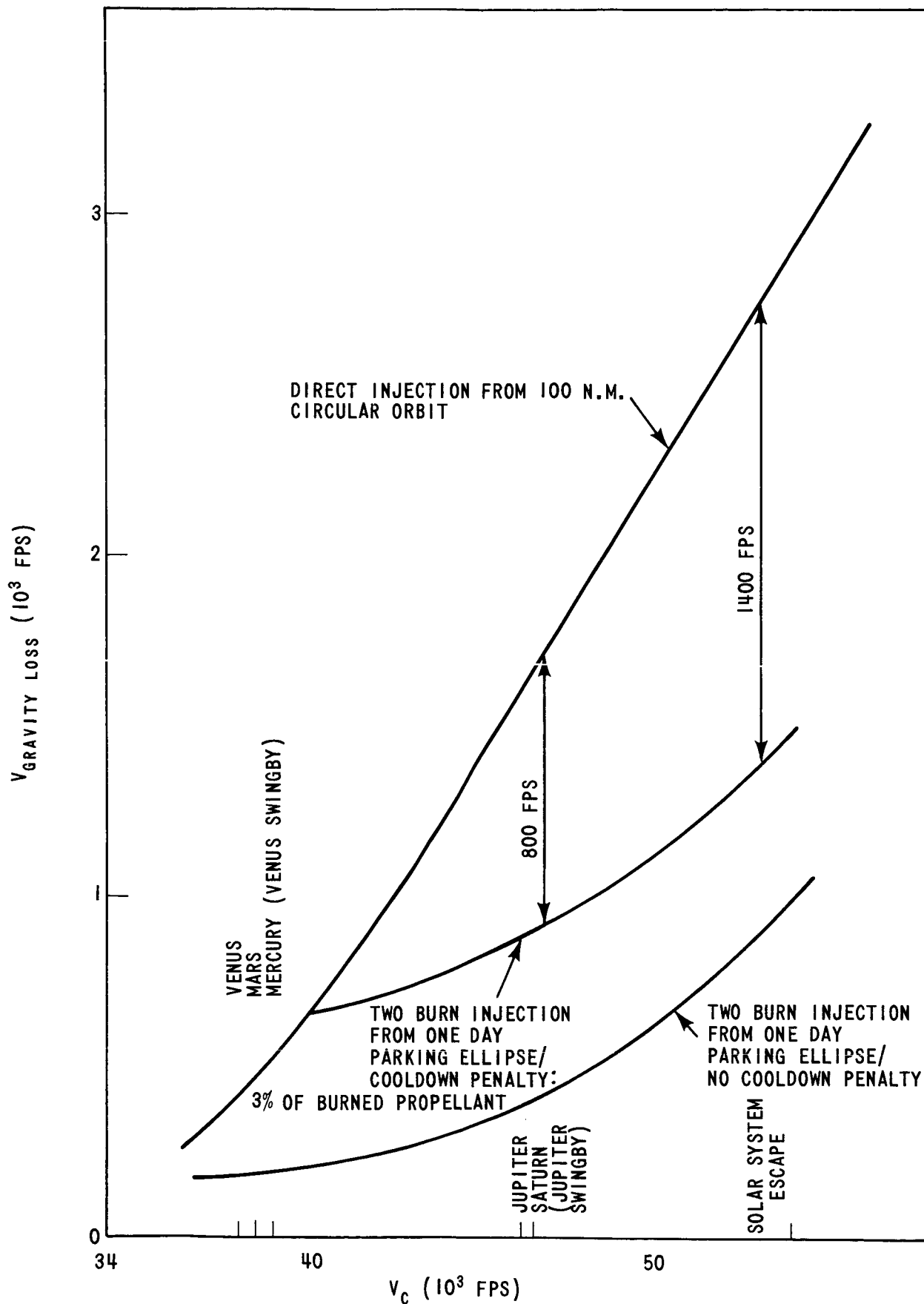


FIGURE 24 - GRAVITY LOSS REDUCTION UTILIZING MULTIPLE BURN INJECTION